Rasmussen

Air cooling of a sloted gas turbine blade.

Thesis
R23

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AIR COOLING OF A
SLOTTED GAS TURBINE BLADE

A Thesis

Submitted to the Graduate Faculty of the

University of Minnesota

by

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in

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SUMMARY

The object of this experiment was to determine the amount of cooling obtained by air cooling a gas turbine blade with slotted cooling air exits, and placed in a flow of hot exhaust gas from a jet burner. The blade was an experimental construction such that both internal cooling and external cooling (from a boundary layer) would result from the cooling air flow.

This test was inspired by previous work in the field of gas turbines in which it has been shown that an increase in efficiency is to be gained at higher operating temperatures. Since metals in present use cannot withstand these temperatures and no new metals or high strength ceramics are in immediate prospect, the trend of endeavour has been toward methods of cooling present blading in order to be able to obtain these desirable high inlet temperatures.

The results of this experiment show that for operating temperatures from about 900° F. to 1600° F., good cooling can be obtained with the configuration used and with relatively low cooling air flow. Using an air

flow equivalent to about 2% of burner inlet air in a J-33 jet engine, an average blade temperature reduction of 420° F. to 620° F. could be expected for the above gas temperatures respectively. The internal cooling was found to be the most effective, however, the boundary layer was seen to be effective near exit from the blade slot and increasingly effective with increased air flow. A trailing edge slotted air exit was found to be effective in reducing trailing edge temperature. Overall cooling effectiveness was found to increase with increased gas temperature.

This test was done in the Main Engines Laboratory of the Mechanical Engineering Department at the University of Minnesota, Minneapolis, Minnesota.

INTRODUCTION

A review of the theory of gas turbines shows that the efficiency of such engines is proportional to turbine inlet temperatures. Consequently, by increasing the inlet temperature of the turbine, a very desirable increase in specific output and decrease in specific fuel consumption can be realized. For a specific engine this can only be realized, however, by a corresponding increase in pressure ratios and R.P.M. In any case it has been found highly advantageous to find means for increasing gas temperature to the turbine without causing serious damage to the blading. With present blading materials, inlet temperatures are limited to around 1000° F. although higher temperatures have been realized by use of impulse staging and other special means. Since no new high temperature metals or high strength ceramics are in immediate prospect, the present line of research has been toward methods of turbine blade cooling. Besides permitting higher inlet temperatures, blade cooling could improve blade life and, for present inlet temperatures, allow use of less strategic or expensive metals for their fabrication.

It is felt that air cooling of turbine blades is the most feasable method of cooling since an unlimited supply of cooling medium is constantly at hand. From previous experiments and analyses, it has been shown that losses introduced by use of reasonable amounts of compressor air for cooling are more than compensated for by increased turbine performance (Ref. 1). In some tests this has amounted to 8% or more of total inlet air to the compressor.

Two tests conducted at the University of Minnesota (Ref. 2 and 3) deal with the cooling of gas turbine blades by means of forming a boundary layer of cool air along the blade surface. Air supplied to a passage inside the blade is ejected through rows of holes in the surface to form the boundary layer. Results of both tests showed that film cooling was most effective near the cooling holes and that the trailing edge of the blade was hardest to cool.

Other methods of blade cooling which have been tried are water cooling by natural convection (Ref. 4), sweat cooling (Ref. 5), and ceramic sleeve cooling (Ref. 6).

Tests have been conducted by the NACA to determine the effects of heat on blades and the critical areas which require cooling or temperature reduotion. In the foregoing tests it was determined that at high temperatures, the leading edge of the blades developed small cracks whioh lead to ultimate failure. Stress measurements at high temperature revealed that the critical area of the blade was approximately at one third the distance from root to tip (Ref. 7).

The basis of this experiment is the use of air under pressure to cool a gas turbine blade with several internal air passages leading to slotted exits. The air thus provides a degree of internal cooling which has been found effective (Ref. 1) and upon exit from a leading edge slot forms a boundary layer of air between the blade and the hot gases. In this manner it was hoped to improve upon previous cooling methods and to determine a comparison between boundary layer cooling and internal cooling.

TEST EQUIPMENT

A schematic diagram of test equipment is shown in Fig. 1. Fig.'s 2, 3, 4, and 5 are photographs of various parts of the equipment used.

The test blade (Fig. 3 and 6) was constructed of mild steel. Three holes were drilled the length of the blade as cooling air passages and joined internally at the blade tip. Slots .01 inches wide were milled lengthwise near the leading and trailing edges and into two of the air passages. The forward slot was constructed in such a manner that cooling air would exit along and parallel to the surface of the blade. The rear slot split the trailing edge to insure cooling air to this thin critical area. Thus air entering the center passage would flow to the outer passages and thence out the slots (Fig. 6). Holes were drilled along the root periphery at points 1-7 as indicated in the blade sketch. Seven thermocouples were placed in these holes to a depth of a little over one third distance from blade root to tip. These thermocouples were used to measure the blade temperature and were connected to a Brown Recording Potentiometer.

Two additional blades were constructed to the same shape but without cooling arrangements. These were placed in the test section to form a cascade with the test blade in the center. Thermocouples were placed similarly in the leading edge of these two blades in order to determine whether or not a temperature variation existed across the flow of hot gases.

The test section was constructed of mild steel to fit the three blades evenly spaced in cascade. A suitable adapter was made to permit removal of the test blade (Fig.'s 2 and 3). The curve seen in the test section permits the flow of hot gases to be turned uniformly with the blading. The curved walls are made to inside and outside blade contours simulating the inside and outside surfaces of two more blades in the cascade. Suitable ducting from the test section leads to burner and exhaust. A total head pressure tube, static tube, and a shielded thermocouple were placed $4\frac{1}{2}$ inches upstream from the blading to measure pressures and temperature necessary for flow measurement.

Hot exhaust gases for the test section were sup-

plied by a J33-A-17 Allison Jet Engine burner. The exhaust of this burner was connected directly to the test section. A spark ignited acetylene flame was used to start the burner diesel fuel spray. (Fig. 5)

Air for the burner was supplied by a 7.48-1 gear ratio centrifugal compressor from an Allison V-1710 aircraft engine. This compressor was operated by a direct drive from a Lycoming O-435-T air cooled light tank engine rated at 162 H.P. at 2800 R.P.M. A six inch square edged orifice was placed in the compressor intake line to measure burner air flow. Pressure differential across the orifice was measured by a water manometer at the control panel. A thermocouple at the inlet measured inlet air temperature. (Fig. 5)

Cooling air for the blade was obtained from a compressed air line in the test cell. This air was piped through a Type 5A-25 Fischer and Porter Flowrator which determined the flow rate of cooling air. Pressure and temperature gauges were placed in the line to obtain data necessary for converting readings to weight flow.

Diesel fuel to the burner was supplied by an

electrically driven aircraft fuel pump and weight flow was determined by a Type 5A-60 fuel flowrator placed in the line.

To ventilate the test cell a large exhaust fan run by a 20 H.P. Westinghouse electric motor was used. Since the capacity of this ventilating system was enough to change the air in the cell several times a minute, the burner exhaust gases were released inside the room near the blower inlet.

The control panel outside the test cell contained all engine instruments, pressure gauges, temperature selector and recorder, manometers, etc. necessary for measuring all items of interest as well as all engine and burner controls. (Fig. 4)

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PROCEDURE

Before starting the jet burner the engine and compressor were thoroughly warmed until oil temperatures read above 120° F. The burner was then started by igniting the acetylene jet at about 1000 engine R.P.M. and then cutting in the fuel. A rise in temperature of gases was noted upon igniting the acetylene which served as a good safety check. The fuel pump was then turned on and fuel pressure built up by the fuel control. A further rise in temperature was noted when the fuel ignited. From here on the engine R.P.M. and fuel pressure were increased simultaneously up to about 90% engine rated R.P.M. and the desired temperature. Acetylene gas was turned off when gas temperature exceeded 600° F. as measured on the shielded thermocouple. It was found that by keeping engine R.P.M. constant, the desired range of temperatures for testing could be obtained by varying the fuel flow.

Readings were made of all instruments at temperatures of 800, 1000, 1200, and 1400 degrees Farenheit as measured on the shielded thermocouple placed in the flow of gases. These readings are tabulated in Table I.

Equations and methods of reducing data are given in the section "Formulas and Sample Computations" of this report. Results of data reduction are shown in Table II.

RESULTS AND DISCUSSION

Upon inspection of the test data it is seen that several of the blade temperatures are in excess of what should have been the total temperature of the gases as measured by the shielded thermocouple. This could be partly due to a variation in temperature of the gases across the test section since the thermocouple was relatively close to the top. However, in all probability the inaccuracy is due to a faulty thermocouple or to a poor method of shielding. In any case, the evaluation of the exact gas temperature is not necessary to the objectives of this report. The temperature, however, serves as a good reference and is indicated throughout the following pages as reference temperature, T_R.

Results are shown graphically in Fig.'s 8 through 18. By examination of the first set of curves (Fig.'s 8 to 11) it is seen that the temperature at the leading edge (station 1) of the blade is in general much higher than at other stations and the air cooling less effective. This was predicted since the thermocouple hole was constructed so close to the leading edge that it almost broke

through. This leaves only a very thin piece of metal to conduct heat away from the point. It is also seen that station 1 is more remote from cooling air than the other stations and should have a higher temperature. At no cooling air flow it is quite possible that this thermocouple gives a good approximation of the gas total temperature.

Fig. 7 serves as a check on consistency and reproducibility of results. Here is plotted the various station temperatures for each run at no cooling air flow. The plotted points are connected by straight lines since a curve through them would be meaningless. It is seen that the general shape so formed is consistent for each test as compared with the others except for station 3 at the higher reference temperature. This run was checked at a slightly lower gas temperature and the same general pattern was repeated as shown. Station 8 in one of the uncooled blades seems to vary from station 9 in the other, indicating a slight variation of gas temperature across the section.

Fig.'s 8-11 definitely show the boundary layer

of cooling air to be effective at station 2. However, as cooling air flow decreases to about 0.4 lb/min the effectiveness of the boundary layer decreases. This is probably not as nearly due to the breaking up of the boundary layer at the lower air flow as it is due to the cooling air reaching a higher temperature before reaching station 2. This is probably also the reason why station 3 apparently receives no benefit from the boundary layer as seen by comparing it with station 6 and 7.

Stations 6 and 7 which have no boundary layer show that the internal cooling is quite effective and apparently evenly distributed to all stations except 1. The results at stations 4 and 5 indicate that the trailing edge slot is very effective.

Fig.'s 12-17 indicate little difference in the amount of blade cooling for gas reference temperatures of 800 and 1000° F. It appears that cooling effectiveness may reach a minimum at about 1000° F. and become slightly unstable at lower temperatures.

Highest temperature reduction was 510° F. at station 2 with maximum air flow.

These results may be approximately compared with an actual jet engine assumed to be operating at the same air flow and gas temperatures per burner. Since a J-33 jet burner was used for this test the comparison here will be for that engine.

The J-33 engine has 54 blades supplied by 14 burners each blade having approximately 15 square inches of area. This makes 3.86 blades per burner of about 58 square inches of area to be cooled per burner. The area of the test blade is 33.8 square inches. Maximum cooling air flow amounted to .71% of burner air flow for the test. By comparing this with the J-33 engine we would have to use

$$\frac{58}{33.8} \times .71 = 1.2\%$$

of burner air to obtain equivalent cooling at the same operating conditions. If 2% of burner air were to be available for cooling it would be necessary to use about 1.59 lb/min in this test to obtain the equivalent data. Since limitations of test instruments prevented using this amount of cooling air, the equivalent amount of cooling can only be approximated. Fig. 18 shows a plot of average blade cooling for each gas temperature. Assuming an ap-

proximately linear relation for higher air flows, the upper and lower curves are extended to a cooling air flow of 1.59 lb/min. Here it is seen that the average blade cooling would lie approximately in the range of from 420 to 620 °F. and would correspond to 2% of burner air flow for the J-33 engine operating at the conditions of the test.

The above comparison is of course approximate since scale effects would have to be taken into consideration in a more accurate comparison.

From Ref. (11) we can compare two geometrically similar blades from the equations below:

$$q = h_o A(t_1 - t_2)$$

$$h_o = .055 \frac{k}{L} \left(\frac{LV}{\mu}\right)^{.75}$$

$$\frac{L_1}{L_2} = \left(\frac{A_1}{A_2}\right)^{\frac{1}{2}}$$

or

$$\frac{q_1}{q_2} = \left(\frac{A_1}{A_2}\right)^{.875} \frac{\Delta T_1}{\Delta T_2}$$

For the areas of 15 and 33.8 square inches assuming

$$\Delta T_1 = \Delta T_2$$

$$q_1 = 2.06\, q_2$$

and for heat flow per unit area the large blade would have .0295 BTU/hr in^2 and the smaller .0324 BTU/hr in^2. In other words, slightly more cooling would be expected from a smaller geometrically similar blade. Therefore, for the J-33 blades it is probable that the cooling temperatures obtained on a straight area ratio are less than actual, assuming actual results to agree fairly closely to those of a geometrically similar blade.

CONCLUSIONS AND RECOMMENDATIONS

From this analysis the following conclusions were reached:

(1) This method of blade cooling gave relatively good overall cooling in a range of gas temperatures from about 900° F. to 1600° F. By using .71% of burner air for cooling, it was found that average temperature reduction ranged from 275° F. to 400° F. for lowest and highest gas temperatures respectively. To cool all the blades in a J-33 jet engine by this method, an equivalent amount of burner air of about 1.2% would have to be used in order to obtain about the same cooling.

(2) Results were extrapolated for the average condition and approximately compared with the J-33 jet engine running at the same conditions with the same cooling arrangement and using 2% of burner air for cooling. It was found that for the same gas temperature range, average temperature reduction could be expected to be from 420° F. to 620° F.

(3) The test blade leading edge was found to

have minimum cooling, being farther from cooling air than other points of measurement on the blade.

(4) The boundary layer cooling was found to have its greatest effect at the higher cooling air flows and was effective apparently for only a short distance along the blade surface (less than one inch).

(5) The greater part of the temperature reduction is attributed to internal cooling. However, this could have reduced the effectiveness of the boundary layer since cooling air was undoubtedly ejected from the slots at fairly high temperature.

(6) The trailing edge slot was found to be very effective in reducing the trailing edge temperature.

(7) Cooling effectiveness (temperature reduction) for the same cooling air flow was noted to increase with gas temperature.

In the event of further experiment or use of this test blade, it is recommended that a more suitable method of gas temperature measurement than the shielded thermocouple be used, that the blade leading edge be cut

down or some modification be applied to cool it better, and that further measurement of the boundary layer effectiveness be made with particular note to the temperature of the cooling air ejected from the slot.

FORMULAS AND SAMPLE COMPUTATIONS

1. Reduction of cooling air data to find weight flow of cooling air. Flowrator calibrated at 14.7 psia and 100° F.

$$W_A = .437\ Q_{A_{read}} \times \frac{(P_A)^{\frac{1}{2}}}{(T_A)^{\frac{1}{2}}}$$

(see Ref. 8)

2. Computation of Mach Number (Ref. 9)

Since static pressure and total pressure were read on manometers attached to the test section, mach number can be determined from the one dimensional compressible flow equation:

$$\frac{p}{p_o} = \left(1 + \frac{r - 1}{2} M^2\right)^{\frac{r - 1}{r}}$$

3. Burner air flow (Ref. 10)

$$w = 2.52 \frac{(p\ x_1 h_w)^{\frac{1}{2}}}{T^{\frac{1}{2}}}$$

NOMENCLATURE

h_w	pressure differential across burner inlet orifice (inches of water)
M	Mach Number
p	test section static pressure (inches Hg.)
p_A	cooling air pressure (psia)
p_o	test section total pressure (inches Hg.)
Q_A	volume flow of cooling air (ft^3/min)
T_1 - T_7	test blade temperatures (°F.)
T_8 - T_9	leading edge temperatures of uncooled blades (°F.)
T_{11}	burner inlet air temperature (°F.)
T_{12}	cooling air temperature (°F.)
T_R	gas reference temperature (°F.)
W_A	weight flow of cooling air (lb/min)
W_f	weight flow of fuel (lb/hr)
w	weight flow of burner air (lb/hr)

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APPENDIX

TABLE I

ORIGINAL DATA SHEET

	$T_R(T_{10})$	P_A	Q_A	P_O	P_S	W_f	h_w	T_1	Diff	T_2	Diff	T_3	Diff
	800	40.4	8.2	32.6	-.2	95	15.4	780	135	575	345	580	270
	800	30.4	6.1	32.7	-.2	95	15.4	820	95	660	260	660	190
	800	22.4	4.2	32.7	-.2	95	15.4	855	60	785	135	730	120
	800	16.4	2.36	32.7	-.2	95	15.4	880	35	825	95	755	95
	800	0	0	32.7	-.2	95	15.4	915		920		850	
	1000	40.4	8.2	33.4	-.2	128	15.4	1010	120	760	360	770	260
	1000	30.4	6.8	33.4	-.2	126	15.4	1045	85	865	255	850	180
	1000	22.4	4.2	33.4	-.2	126	15.4	1075	55	960	160	910	120
	1000	16.4	2.42	33.4	-.2	126	15.4	1110	20	1050	70	980	50
	1000	0	0	33.4	-.2	126	15.4	1130		1120		1030	
	1210	40.4	8.2	34.0	-.2	158	15.4	1225	130	920	440	925	330
	1210	30.4	6.0	34.0	-.2	158	15.4	1255	100	1010	350	1000	255
	1200	22.4	4.25	34.0	-.2	159	15.4	1290	65	1160	200	1090	165
	1210	16.4	2.48	34.0	-.2	159	15.4	1330	25	1280	80	1160	95
	1210	0	0	34.0	-.2	159	15.4	1355		1360		1255	
	1400	40.4	8.5	33.8	-.1	169	15.4	1390	170	990	510	1050	440
	1400	30.4	6.6	33.9	-.1	169	15.4	1430	130	1120	380	1170	320
	1400	22.4	4.1	33.9	-.1	169	15.4	1470	90	1270	230	1275	215
	1395	16.4	2.3	33.9	-.1	169	15.4	1520	40	1390	110	1360	130
	1410	0	0	33.9	-.1	169	15.4	1560		1500		1490	
Check	1360	0	0	33.9	-.1	166	15.4	1525		1470		1450	

Ambient Temp. - 87° F. Barometer - 29.23" Hg.

TABLE I (cont.)

ORIGINAL DATA SHEET

T_4	Diff	T_5	Diff	T_6	Diff	T_7	Diff	T_8	T_9	T_{10}	T_{11}	Average Diff.
610	290	605	275	630	275	605	305	850	840	90	90	271
680	220	665	215	700	205	680	230	850	845	90	90	202
750	150	730	150	770	135	770	140	855	845	90	90	127
795	105	775	105	810	95	840	70	855	850	90	90	86
900		880		905		910		865	855	95	95	
800	300	800	270	830	270	800	320	1070	1080	95	85	271
880	220	865	205	910	190	890	230	1070	1085	95	85	195
950	150	920	150	970	130	980	140	1070	1085	95	85	129
1000	100	970	100	1025	75	1060	60	1070	1090	95	90	68
1100		1070		1100		1120		1070	1090	95	90	
970	360	960	330	1010	320	970	380	1245	1310	95	85	327
1050	280	1030	260	1080	250	1060	290	1255	1330	95	85	255
1130	200	1100	190	1160	170	1170	180	1260	1330	95	85	167
1200	130	1165	125	1235	95	1280	70	1265	1335	95	90	89
1330		1290		1330		1350		1270	1340	95	90	
1060	435	1040	390	1070	400	1050	440	1355	1430	105	95	398
1170	325	1130	300	1175	295	1180	310	1380	1460	105	95	294
1270	225	1210	220	1270	200	1280	210	1390	1460	110	100	200
1335	160	1280	150	1350	120	1395	95	1395	1430	110	100	115
1495		1430		1470		1490		1410	1480	110	100	
1460		1390		1430		1460		1390	1445	110	100	

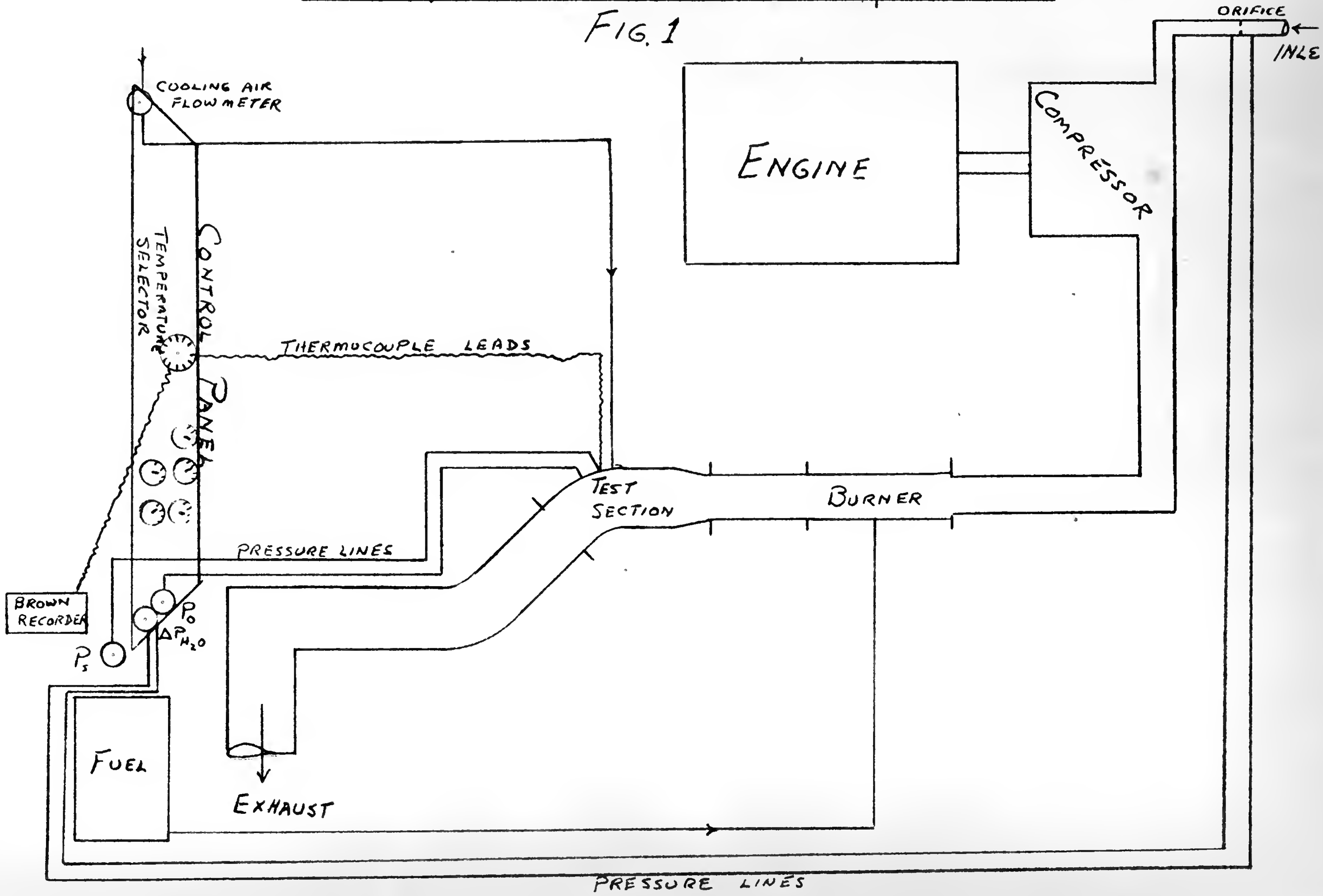
TABLE II

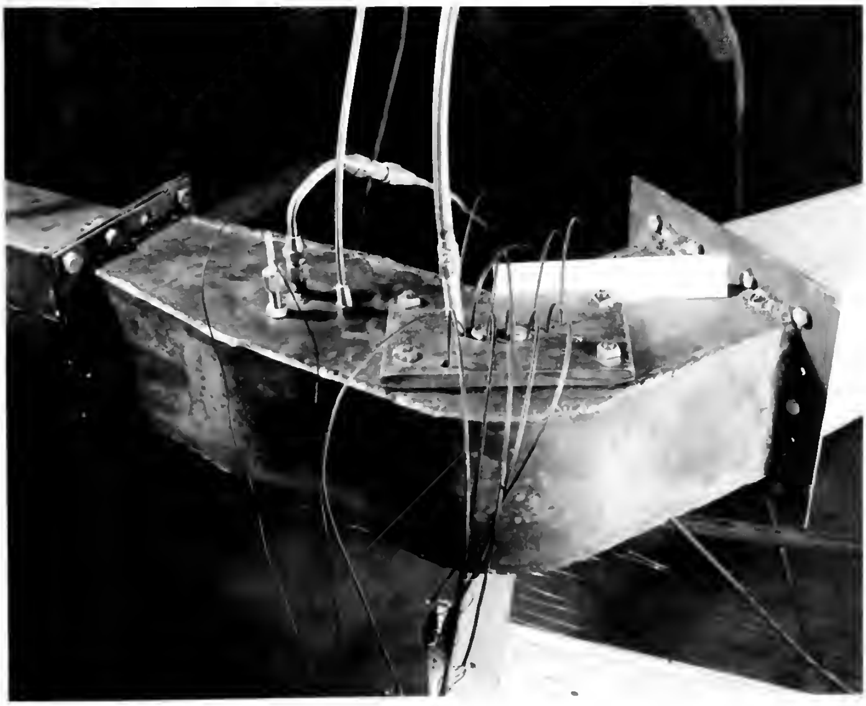
REDUCED DATE

T_R	W_A	M	W_f	w	W_f/w
800	.973	.43	95	8200	.0116
800	.625		95		
800	.367		95		
800	.176		95		
800	0		95		
1000	.973	.47	128	8200	.0156
1000	.695		126		.0154
1000	.370		126		
1000	.181		126		
1000	0		126		
1210	.973	.50	158	8200	.0193
1210	.618		158		
1200	.374		159		.0194
1210	.187		159		
1210	0		159		
1400	.962	.49	169	8200	.021
1400	.677		169		
1400	.358		169		
1400	.172		169		
1410	0		169		

SCHEMATIC DIAGRAM OF EQUIPMENT

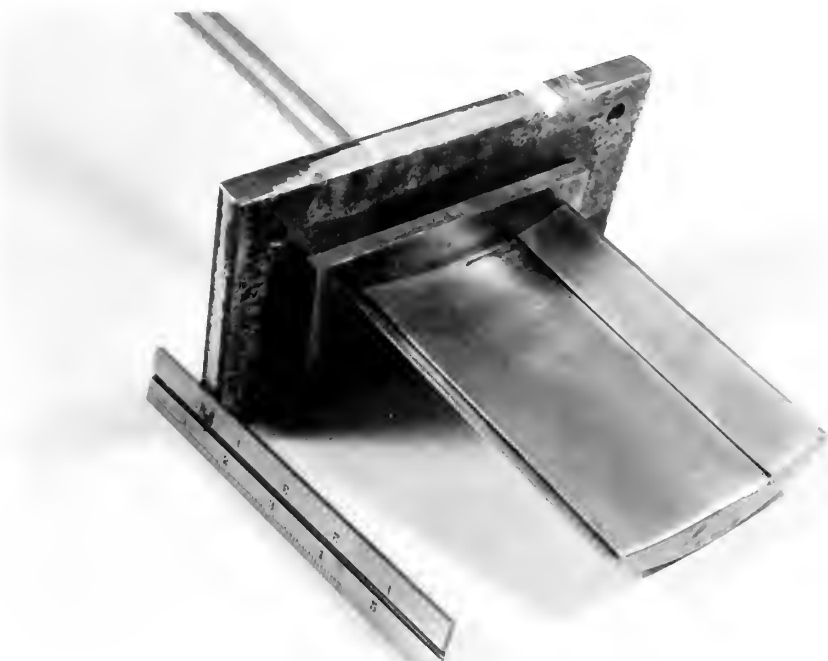
FIG. 1





TEST SECTION

FIG. 2

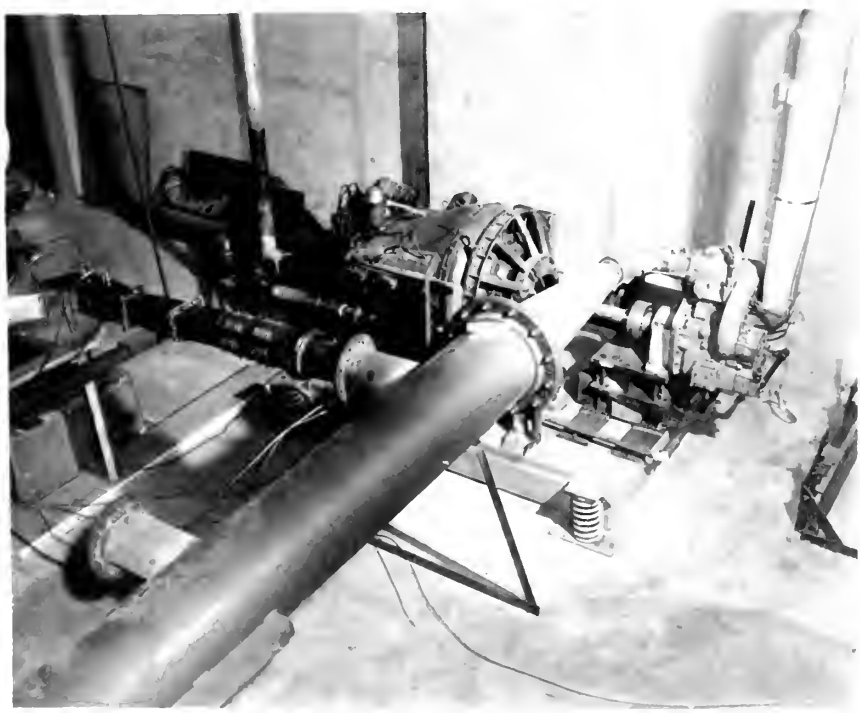


TEST BLADE

FIG. 3

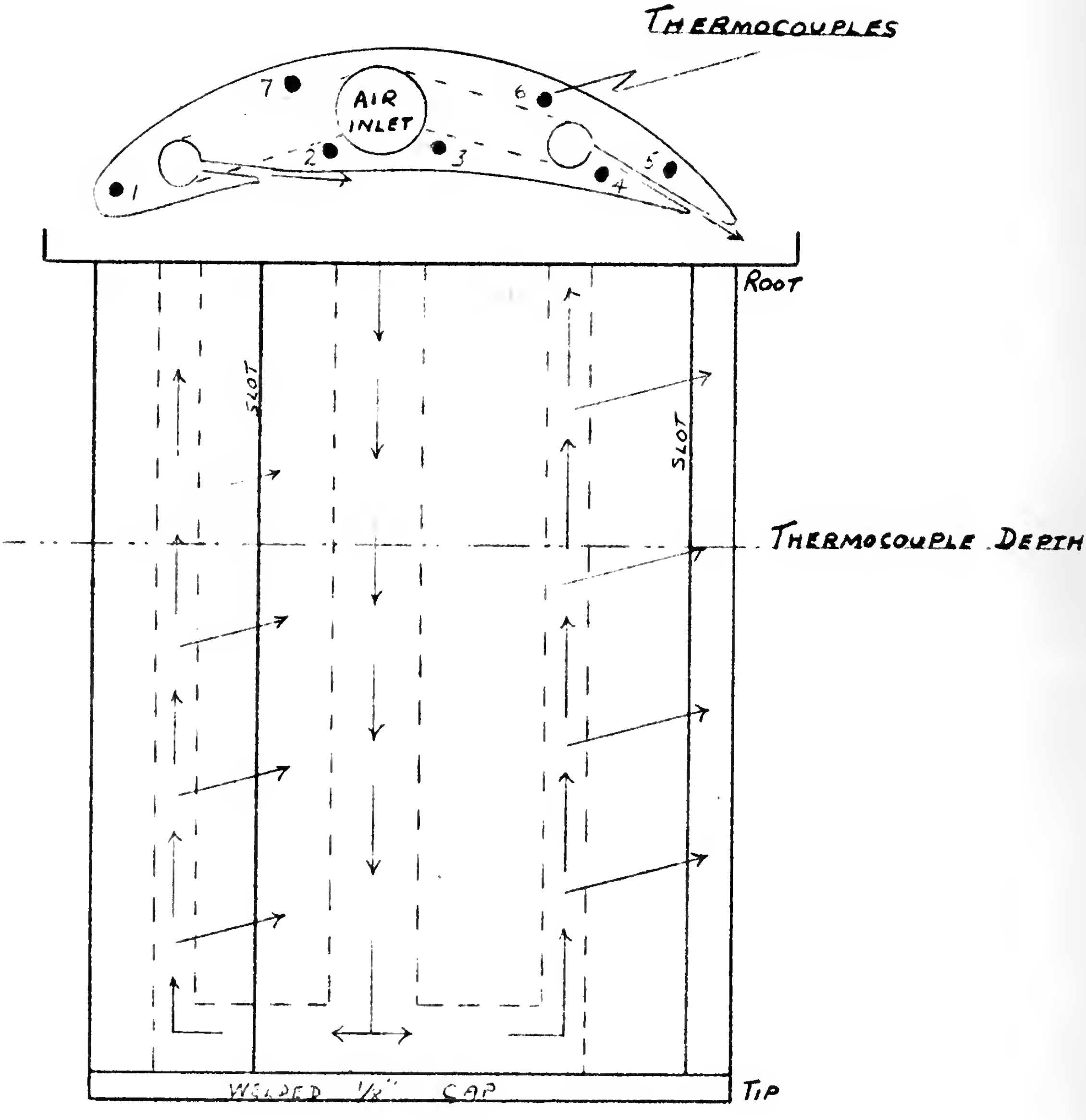
CONTROL PANEL

FIG. 4



TEST CELL

FIG. 5



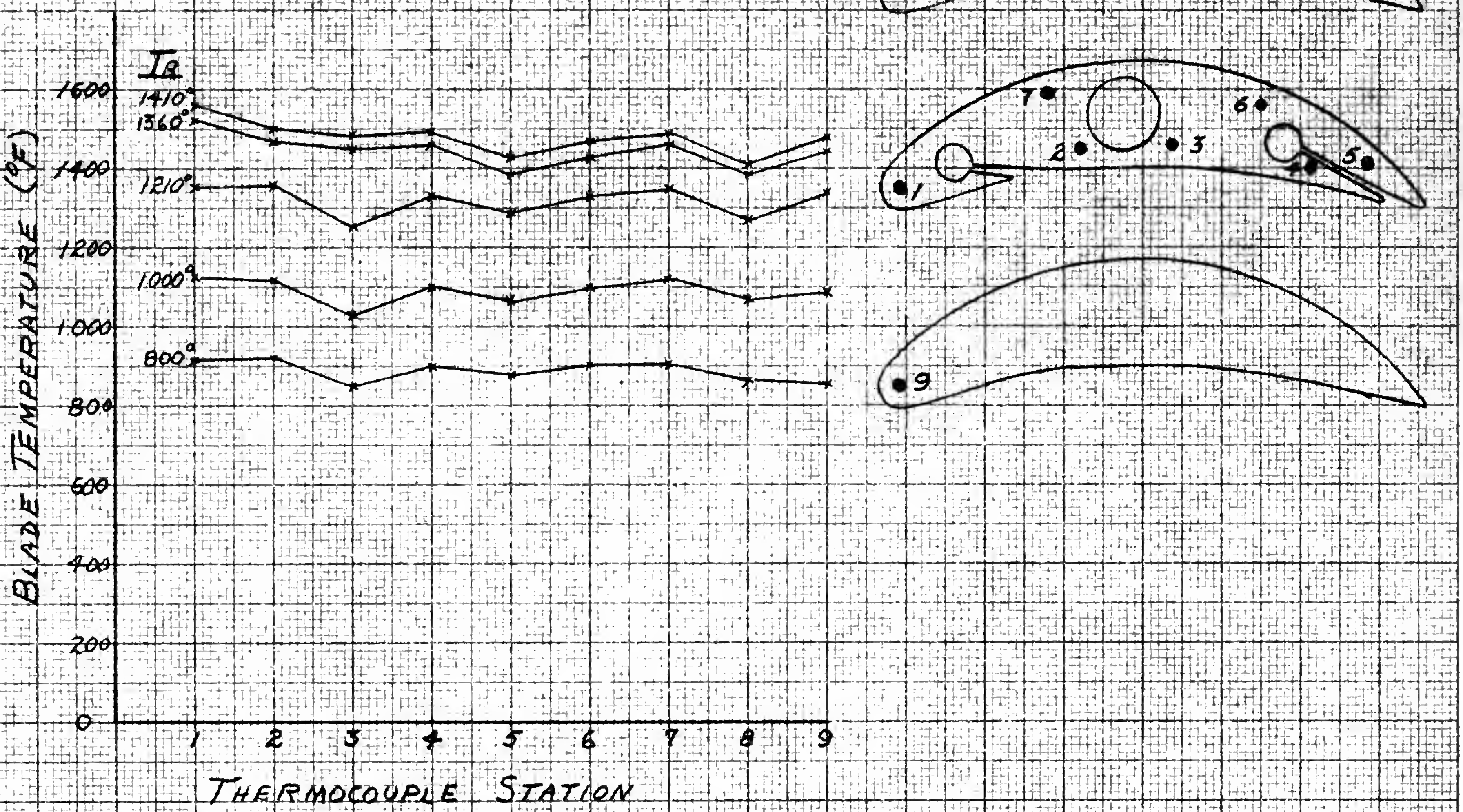
ARROWS INDICATE AIR FLOW

AIR PASSAGES IN BLADE

FIG. 6

BLADE TEMPERATURES WITH NO COOLING AIR FLOW

FIG. 7



BLADE COOLING AT REFERENCE TEMPERATURES

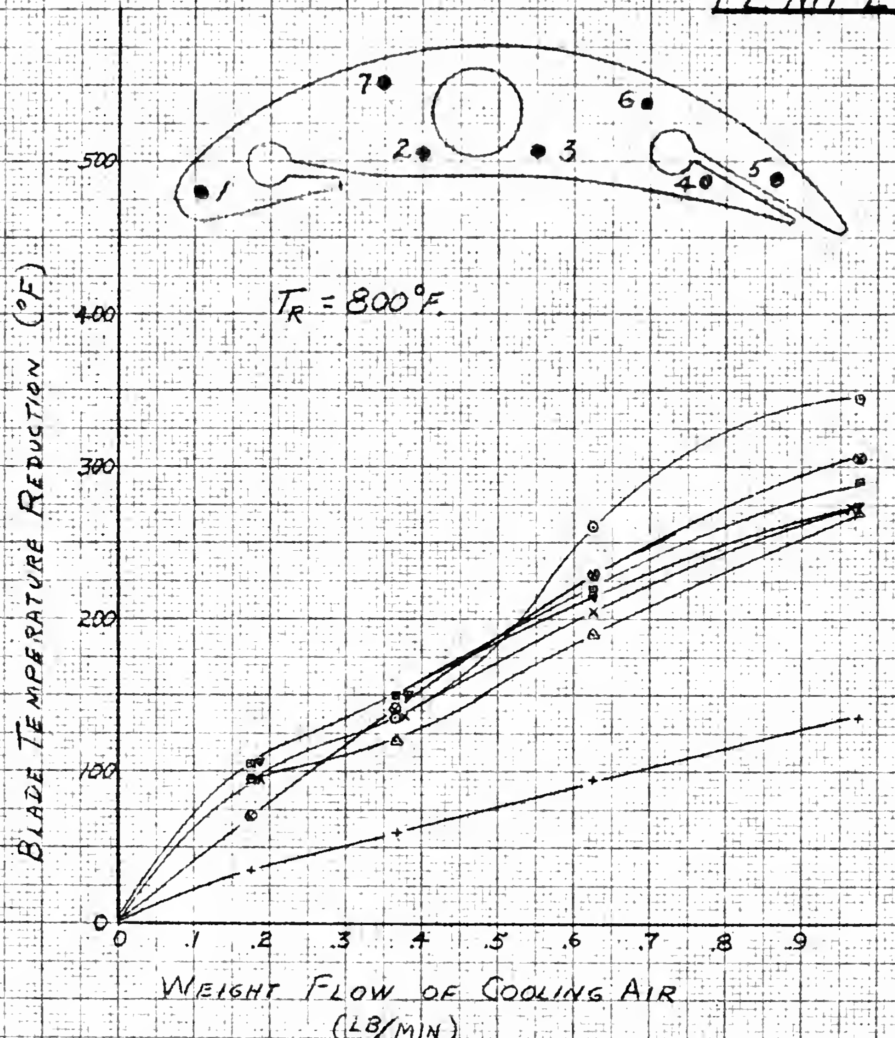


FIG. 8

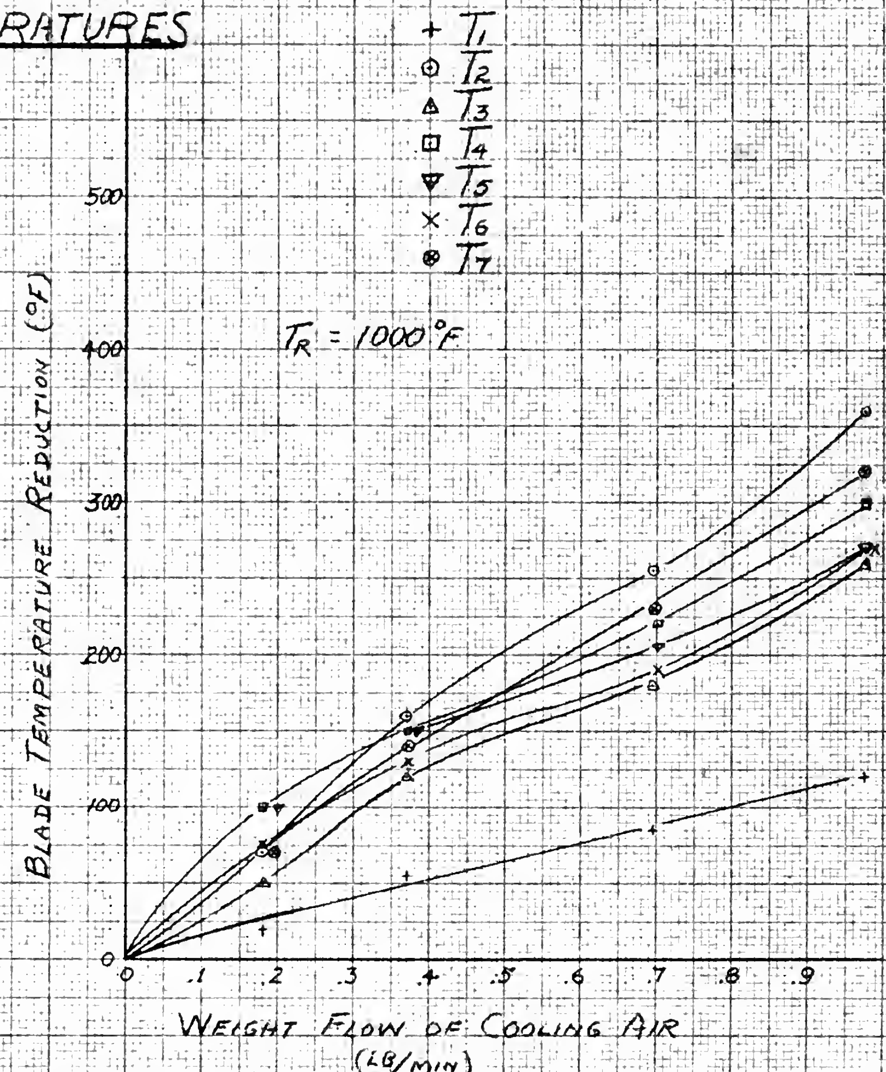


FIG. 9

BLADE COOLING AT REFERENCE TEMPERATURES

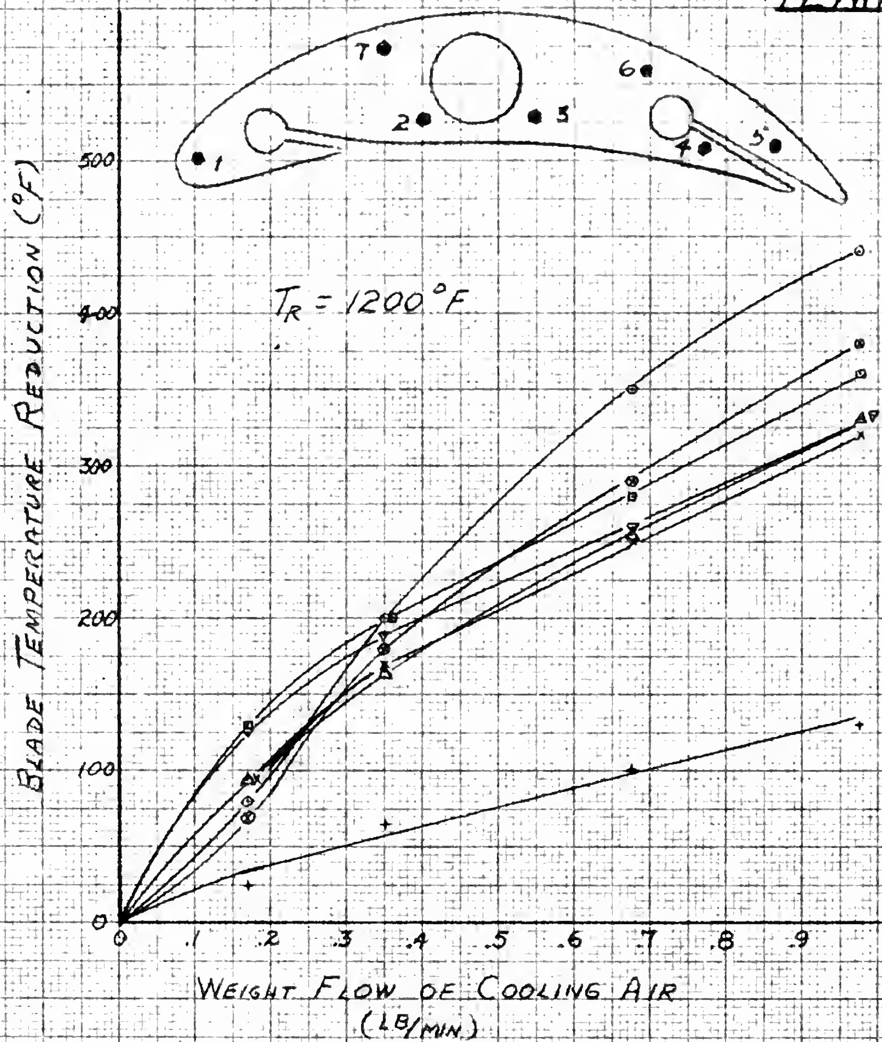


FIG. 10

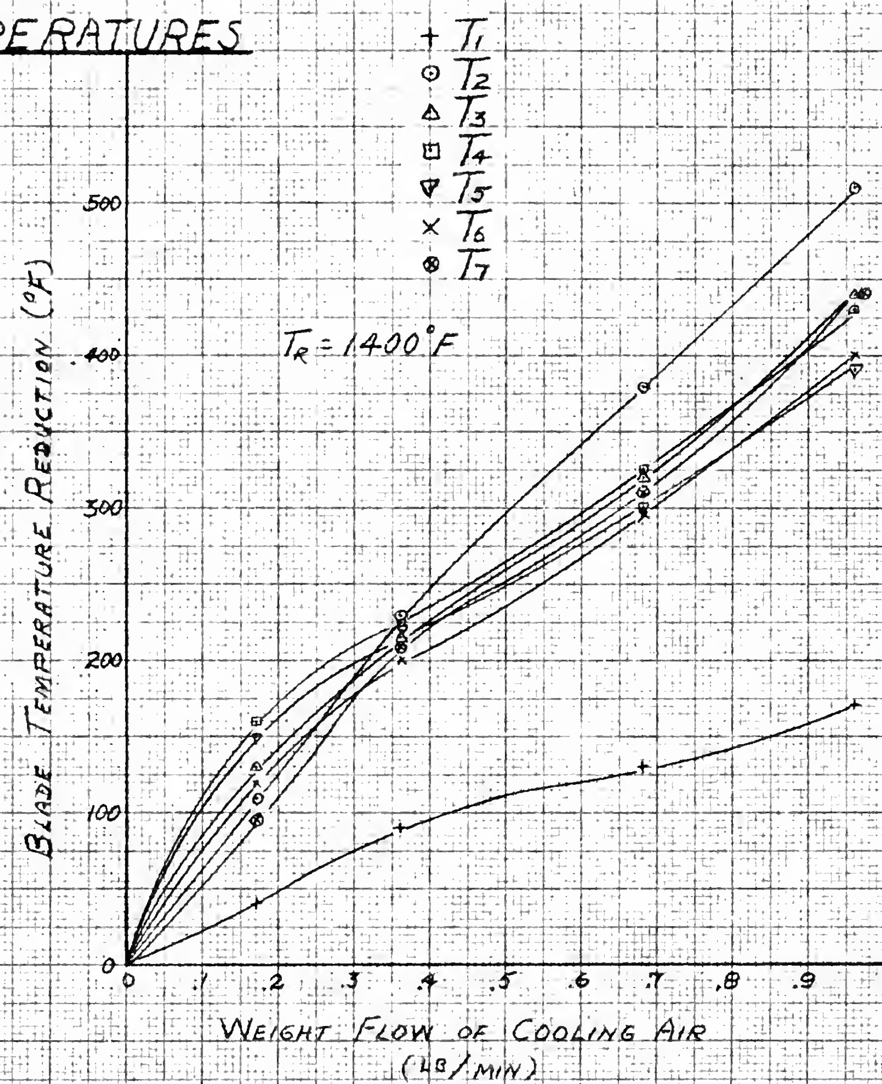


FIG. 11

BLADE COOLING AT EACH STATION

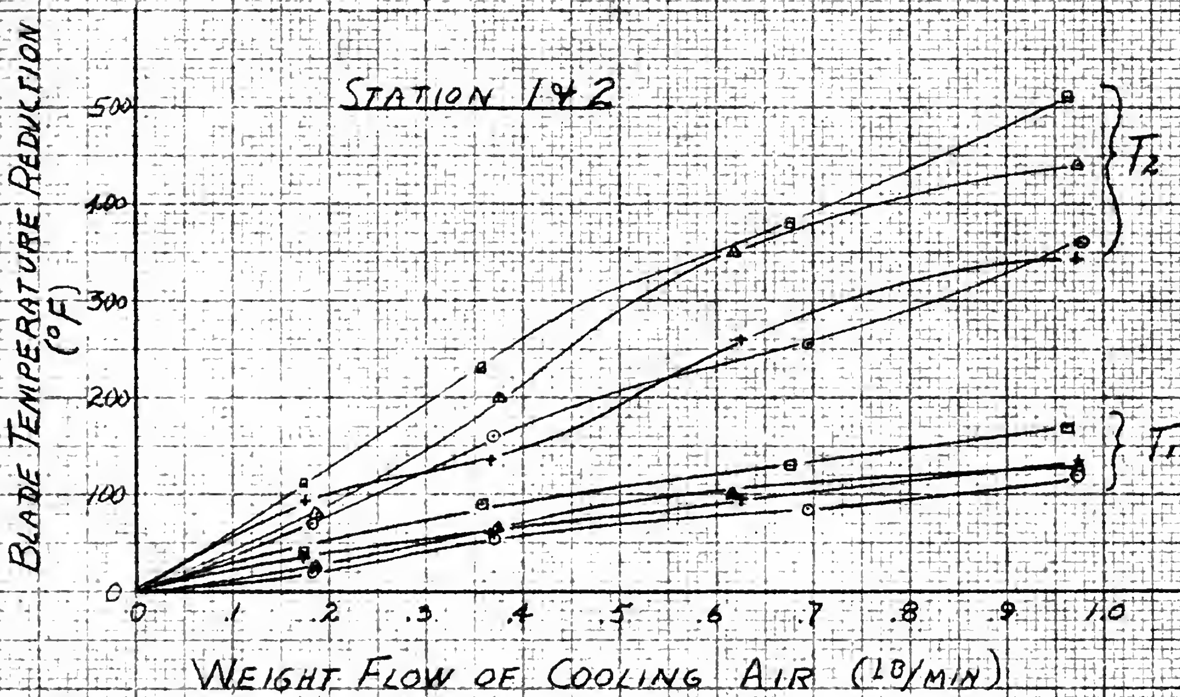
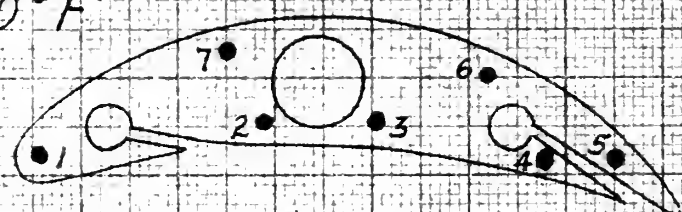


FIG. 12

+ T_R = 800 °F
⊙ T_R = 1000 °F
△ T_R = 1200 °F
⊡ T_R = 1400 °F



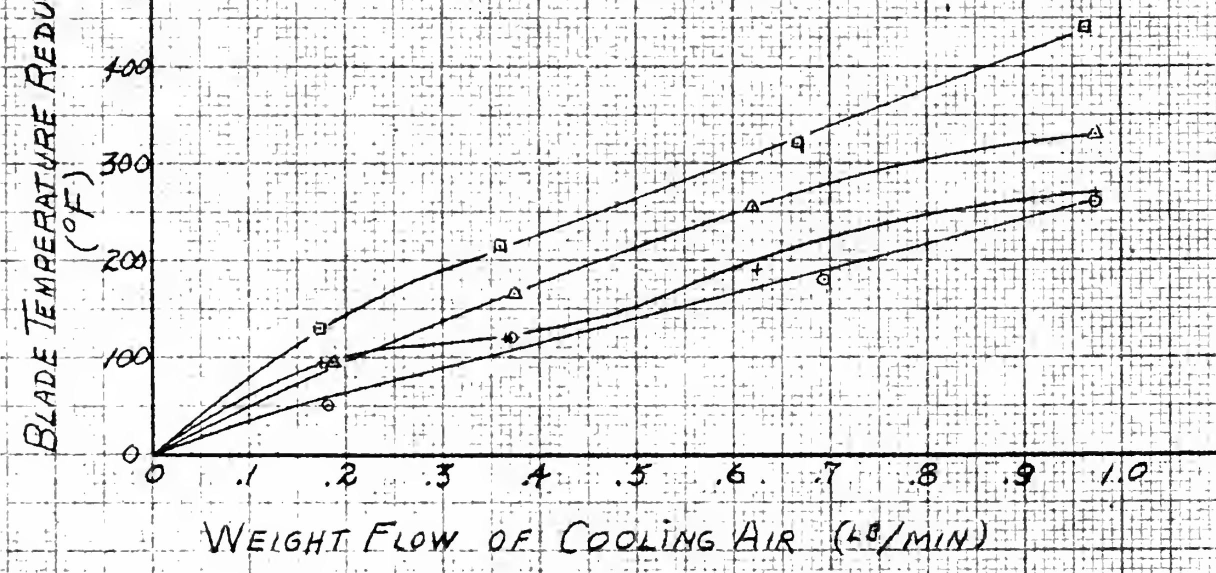


FIG. 13

BLADE COOLING AT EACH STATION

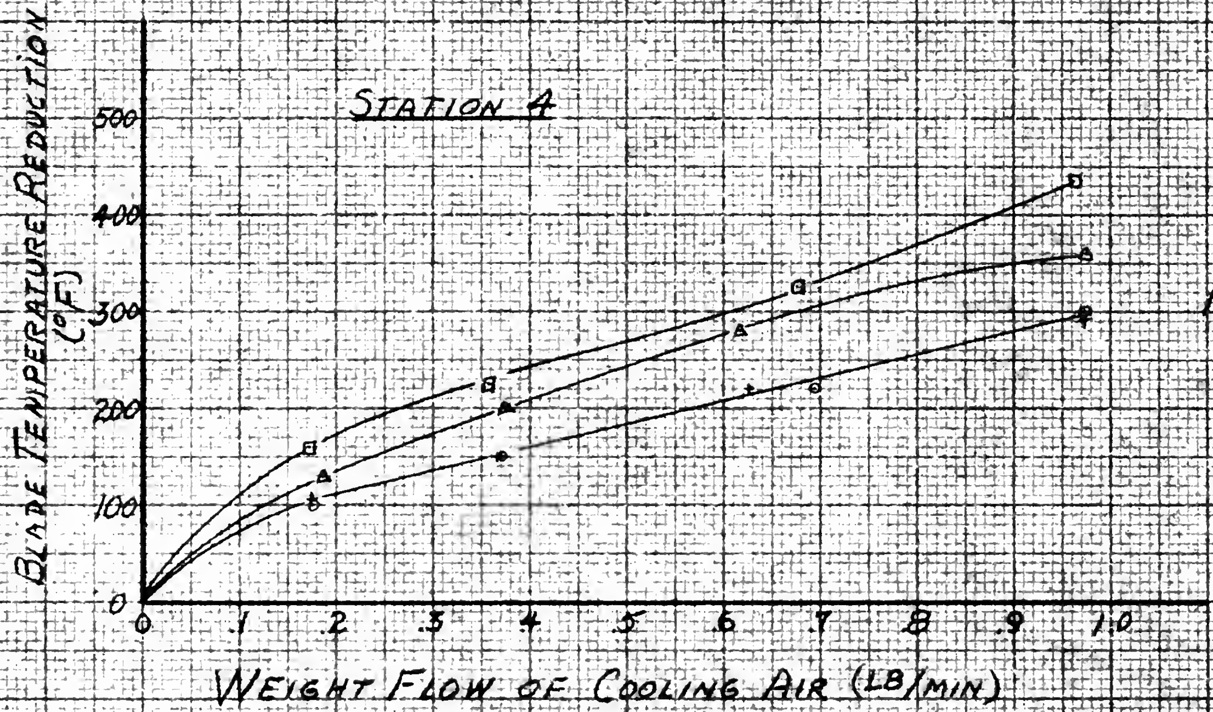
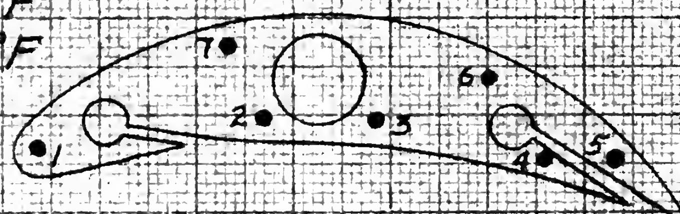


FIG. 14

+ T_R = 800°F
⊙ T_R = 1000°F
△ T_R = 1200°F
⊡ T_R = 1400°F



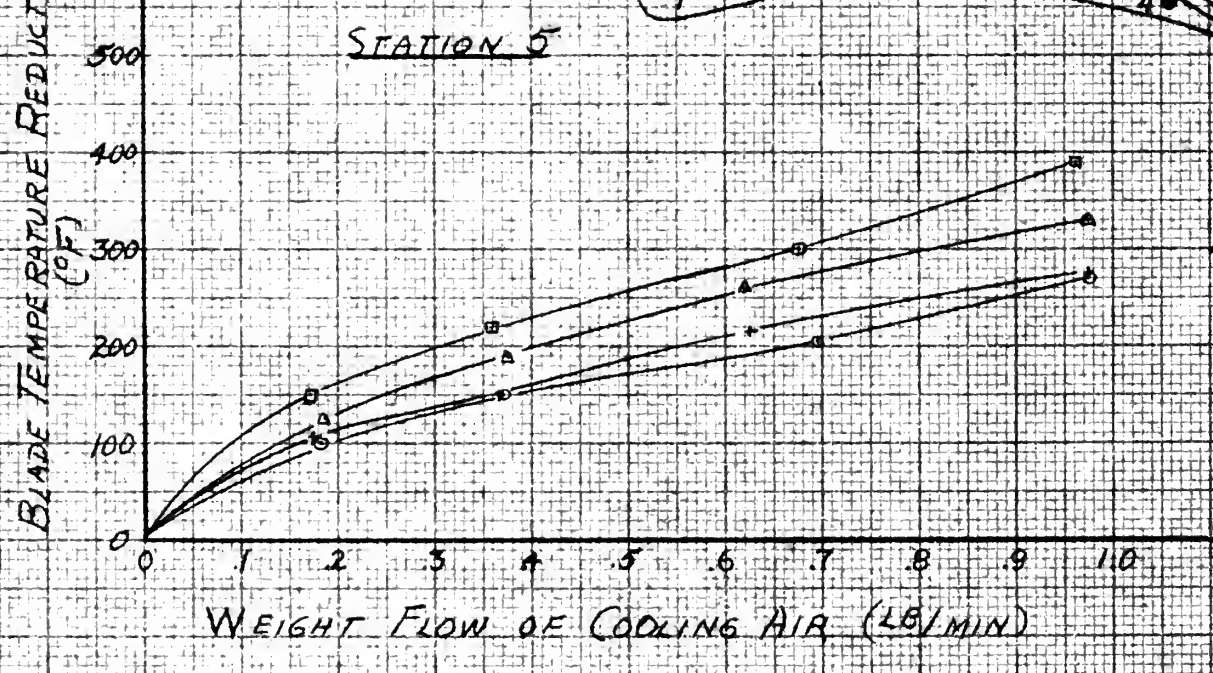


FIG. 15

BLADE COOLING AT EACH STATION

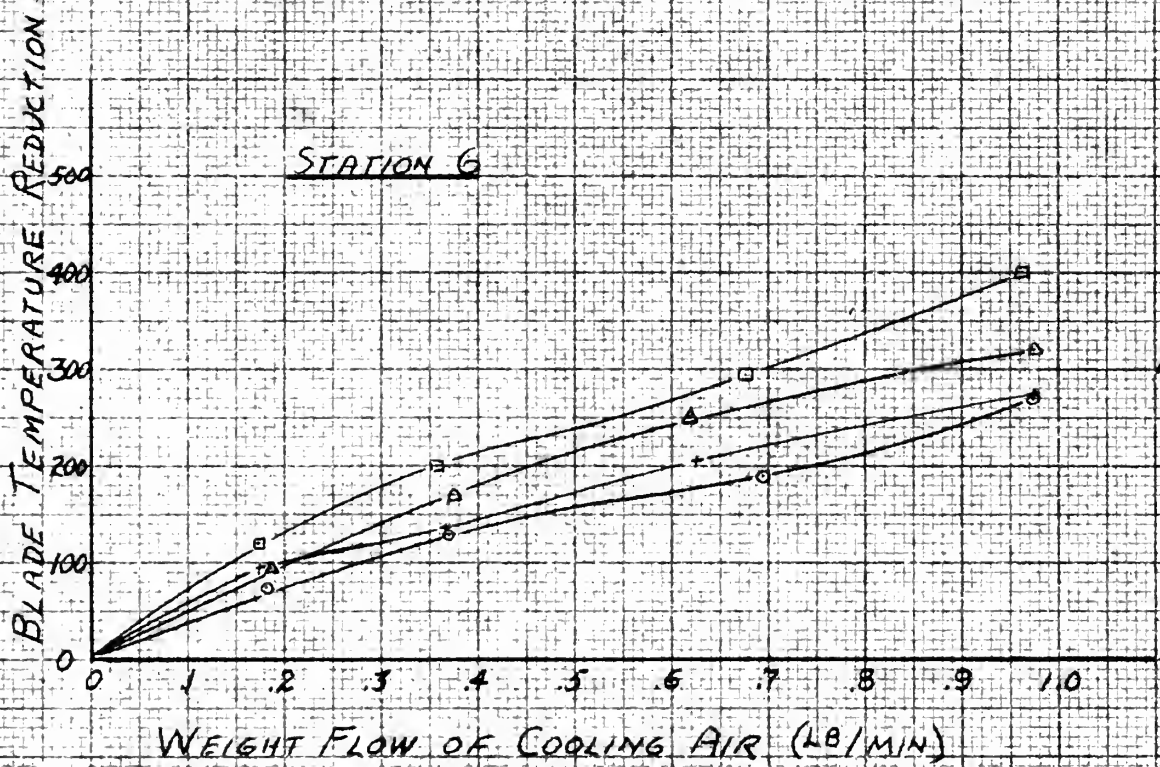
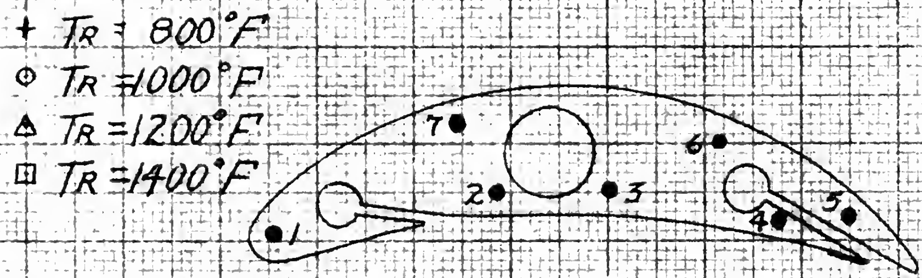


FIG. 16



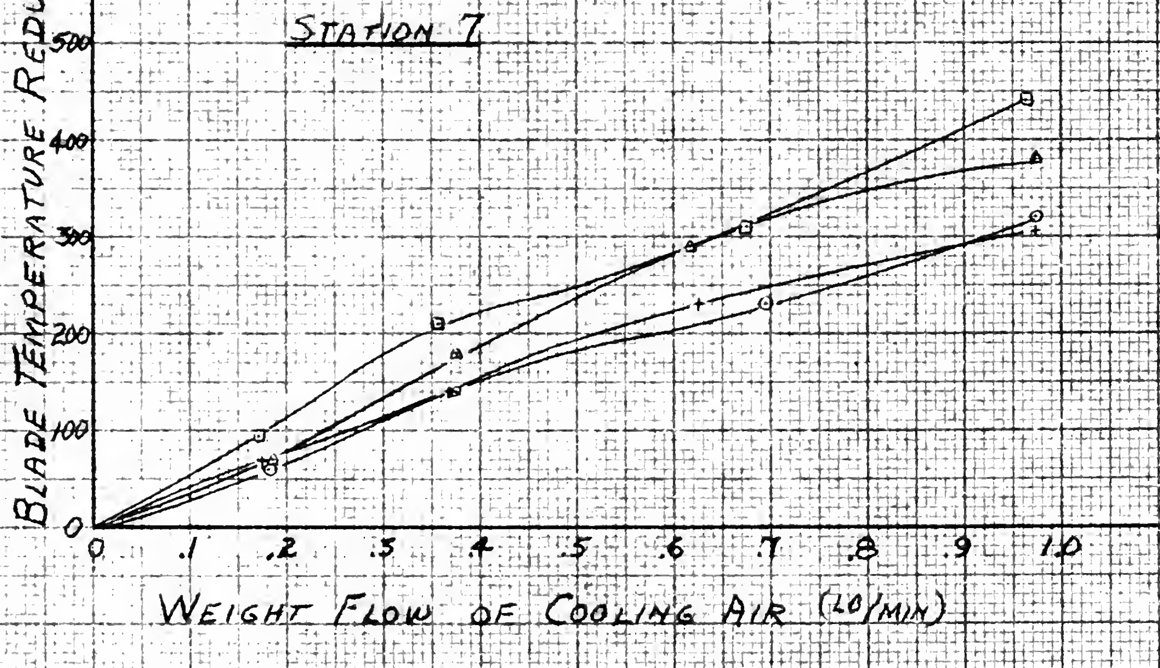


FIG. 17

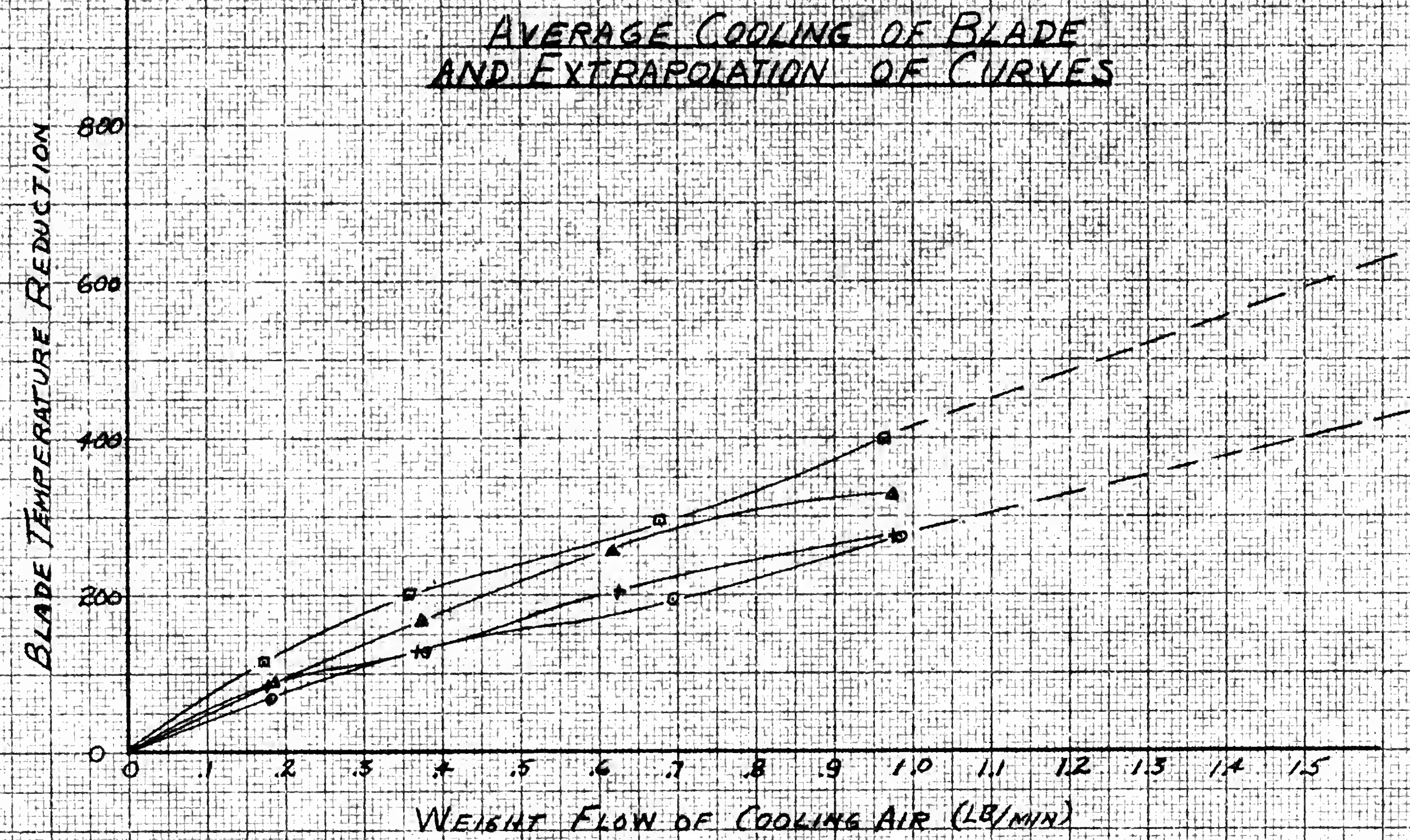

AVERAGE COOLING OF BLADE
AND EXTRAPOLATION OF CURVES
BLADE TEMPERATURE REDUCTION
800
600
400
200
0
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
WEIGHT FLOW OF COOLING AIR (LB/MIN)

FIG. 18

Thesis 16262
R23 Rasmussen
Air cooling of a slotted gas turbine blade.

Thesis 16262
R23 Rasmussen
Air cooling of a slotted gas turbine blade.

thesR23
Air cooling of a slotted gas turbine bla
3 2768 002 05299 5
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